P150500.PDF [Page: 3 of 10]

Composite Repair Issues on the CF-18 Aircraft

A.J. Russell and J.S. Ferguson

Defence Research Establishment Pacific
F.M.O. Victoria, B.C., Canada
V0S 1B0

150500

1. SUMMARY

This paper addresses three separate composite repair issues currently being investigated at the Defence Research Establishment Pacific (DREP) in support of Canada's CF-18 aircraft. First, the problem of skin/core debonding that can occur during elevated temperature bonded repairs of honeycomb sandwich structure is discussed and the results of tests which help to quantify the role played by bondline degradation are presented. Next, the final development phase of an on-going effort to establish a reliable and effective means of repairing delamination damage is reported. In particular, the design of a resin injection device and a series of tests to evaluate its performance are described. Finally, a battle damage repair issue, namely the advantages and disadvantages of cleaning up the damaged composite material found around entry and exit holes created by live fire, is discussed. Test data is presented which compares the different consequences under tensile and compression loading.

2. LIST OF SYMBOLS

d_o	characteristic distance for determining point stress
K_t	stress concentration factor
R	radius of hole
σ_0	un-notched material failure stress
σ_N	notched material failure stress

3. INTRODUCTION

Although they provide significant advantages in specific strength and stiffness compared to traditional metallic designs, composite aircraft components create new challenges for maintenance personnel due to the unique nature of their damage modes. While composites offer the potential for significant savings in life cycle costs due to their excellent fatigue and corrosion resistance, their susceptibility to impact damage, especially in thin sections, requires more careful handling as well as the development of appropriate repair methods. On the CF-18 the most frequently required composite repair is to impact damage of the lightweight honeycomb sandwich structure. These repairs often involve removal of the damaged skin and core, the splicing in of new core material and the adhesive bonding of an external patch. Although these repairs are normally quite effective in restoring the component's performance to pre-damage levels occasional problems have been encountered during elevated temperature curing of the repair adhesive. In particular, sufficient preasure can develop within the cells of the honeycomb to blow the skin from the core. The probable origins of this problem as well as the remedial action that has been taken will be described. In addition, tests carried out in order to gain a better understanding of the precise mechanisms responsible will be presented.

The monolithic carbon/epoxy structure on the CF-18 is also prone to impact. However, the greater skin thicknesses and their correspondingly higher thresholds for impact damage have meant that relatively few repairs have been necessary. Nevertheless, the ability to effectively repair delaminations that exceed a critical size without having to remove the wing skin or resort to a bolted patch repair is considered to be both desirable and cost effective. The topic of delamination repair as it pertains to the carbon/epoxy wing skins of the CF-18 has been discussed in some detail in a previous paper presented at the AGARD specialists meeting on Delamination and Debonding in 1992 [1]. The successful development of an epoxy resin, specially formulated for injection repair of delaminations, was described as well as its ability to completely restore the through-the-thickness laminate properties. More recently, follow-on work has focused on developing the necessary equipment and procedures to inject the resin into delaminations in-situ. These latest efforts will be reported here.

Damage inflicted in the course of combat is by its very nature quite different from that encountered during peacetime. Also, time and logistics constraints during conflict limit the type and extent of repairs that can be implemented. Because of this, Aircraft Battle Damage Repair (ABDR) is generally considered separately from other repair activities. Nevertheless, the key decisions that must be made are the same: does the damage need to be repaired? and, if so, how can it be done? In the case of composites, battle damage often takes the form of jagged edged through-thickness holes surrounded by a region containing delaminations, matrix cracking and plies peeled from the back surface. Since much of this damage can not be seen and since other inspection techniques such as ultrasonic scanning are not amenable to an ABDR environment, repair/no repair decisions must be based on the size of the visible damage and a prior knowledge of the structural significance of damage of this type. The results of tests carried out in order to address this particular ABDR issue will also be presented in this paper.

Paper presented at the 79th Meeting of the AGARD Structures and Materials Panel on "Composite Repair of Military Aircraft Structures" held in Seville, Spain 3-5 October 1994.

P150500.PDF [Page: 4 of 10]

4. BONDED REPAIRS

During the past two or three years there have been several instances where new damage has been introduced into F-18 honeycomb components undergoing bonded repair. The problem has manifested itself in the debonding of extensive sections of carbon/epoxy skin from the core during the elevated temperature curing of repair adhesives. Several factors are believed to have contributed to these failures. The repair adhesive cure temperature of 150°C is only 30°C below the original cure temperature of the American Cyanamid FM-300 skin/core adhesive and 10°C above its glass transition temperature. Existing heating blankets and single thermocouple controllers can result in temperature excursions of up to 10°C above and below the intended cure temperature. Moisture, which has diffused slowly into the carbon/epoxy skin over the life of the aircraft, eventually reaches the bondline where it has two effects. First, it further reduces the elevated temperature strength of the adhesive and second, its vapour pressure can raise the pressure within the core significantly above that of heated air alone. Figure 1 shows the increase in pressure as a function of temperature as well as data from McDonnell Douglas showing the flatwise tension strength of aluminum honeycomb samples bonded

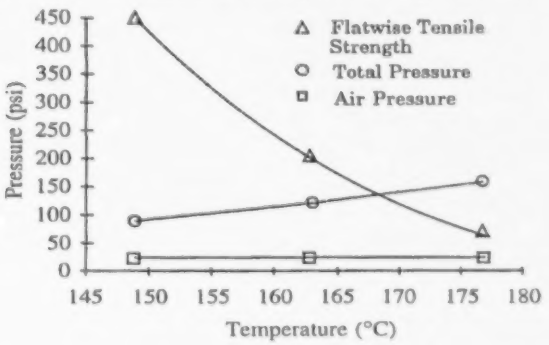

Figure 1 - Bond Strength and Vapor Pressure versus Temperature in Honeycomb Structure.

with FM-300. The total pressure is calculated by adding the air pressure with the vapor pressure at the desired temperature. Clearly the risk of skin/core debonding can be reduced by lowering the repair temperature and by completely drying the skin prior to repair. While the qualification of a new 121°C curing adhesive for F-18 repairs has still to be completed, extending the drying cycle to up to four days and increasing the drying temperature has not in itself eliminated skin/core debonding during repair. Because of this, further investigation of the problem was initiated.

One possible failure mode that was not initially considered is moisture induced degradation of the aluminum/adhesive interface. Such a mechanism is consistent with the latent development of the debonding problem and the lack of any pre-bond surface treatment or application of primer during fabrication of the F-18 carbon/epoxy and aluminum honeycomb sandwich structure. In order to study this possibility further, a number of climbing drum peel specimens, which were representative of the honeycomb structure on the CF-18 were manufactured. Facesheets, made up of three unidirectional plies of AS4/3501-6 carbon/epoxy in a [0/90/0] lay-up, were adhesively bonded with FM-300 film adhesive to 12.5 mm thick 5056 aluminum alloy core. The facesheets were made as thin as possible to reduce the moisturizing time and yet sufficiently strong to carry the load needed to fail the bondline. After sealing the specimen edges with silicone sealant some of the test specimens were moisturized in a 100% relative humidity environment at 67°C. The change in specimen weight, which was monitored periodically during conditioning, is shown in Figure 2. Compared

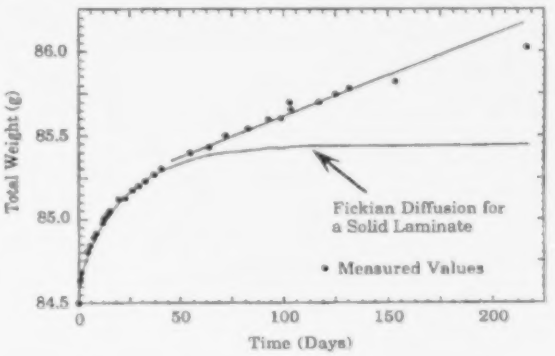

Figure 2 - Weight Gain Curve for Climbing Drum Peel Specimen Conditioned at 100% R.H./60°C.

with normal Fickian diffusion in solid laminates, Figure 2 shows a steady state region (between 50 and 130 days) where the rate of increase in weight is constant indicating that the core is acting as an effective sink for the absorbed moisture.

Mechanical testing was carried out in a similar manner to that described in ASTM D 1781 but at 121°C and 149°C. Prior to testing, the moisturized specimens were dried at 88°C under vacuum for two days followed by two days at 107°C. Even with this extended drying cycle, which duplicates one that has been recommended for F-18 honeycomb components, a small percentage of the absorbed moisture remained. Figure 3 shows a typical load versus displacement plot obtained from one of the climbing drum peel tests. The

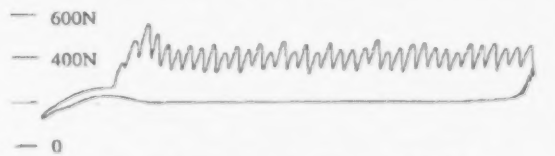

Figure 3 - Load-Displacement Plot for Climbing Drum Peel Test at 149°C dry.

peel strength was determined from the average of the peak loads after subtracting the average load measured during unloading. Data from the initial portion of the curve was discarded. Figure 4 shows the peel strengths measured for each of the different conditions examined. Also shown are the minimum values that must be met during routine quality control testing of

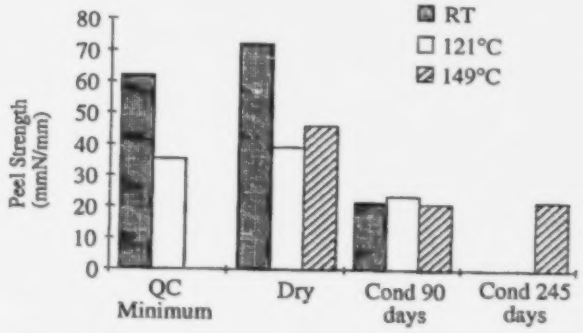

Figure 4 - Climbing Drum Peel Strength of Honeycomb Samples.

FM-300. It is quite evident that the conditioned and dryed specimens have lost a significant amount of their original peel strength. Moreover the peel strength has lost its dependancy on the test temperature. Conditioning beyond 90 days has no further effect on the strength. Examination of the failure surfaces revealed that the failure mode had also changed. While the failure surfaces of the unconditioned specimens showed about 95% cohesive failure with adhesive fillets left bonded to most of the core, the conditioned specimens exhibited approximately 95% adhesive failure with the core pulling cleanly out of the adhesive fillets. This latter mode of failure has been observed on a U.S. Navy F–18 trailing edge flap that had debonded during repair [2].

It has been well established that the durability of aluminum adhesive bonds depends on the stability of the oxide layer [3]. In the absence of appropriate pre-bond surface treatments moisture can transform the oxide to boehmite which has poor adhesion to the aluminum substrate, thereby reducing the bond strength. An important variable that affects the rate of this reaction is the temperature, with significantly longer induction times being required below 50°C [4]. Additional climbing drum peel specimens are currently undergoing conditioning at 30°C to ascertain if similar degradation occurs at that temperature but these have not yet absorbed sufficient moisture to warrrant testing.

There would appear to be two distinct ways of preventing skin core debonding problems during on-aircraft elevated temperature repairs. On the one hand, repair temperatures could be restricted so that the vapour pressure generated is insufficient to exceed the strength of the skin/core bond. Alternatively, drying times could be extended until no moisture remained to raise the pressure above that of the expanded air thus allowing the use of higher temperature curing adhesives. Both approaches have their drawbacks. Lower temperatures generally produce lower hot/wet adhesive properties which may restrict the severity of the damage/strain level combinations that can be repaired. Complete drying of honeycomb skins, which may be up to 30 plies thick, could take several weeks based on the present results. Not only would this adversely affect aircraft availability, the prolonged exposure to higher temperatures is also likely to exacerbate bondline degradation. During the early stages of drying the humidity level in the core may actually increase as the prevailing moisture gradient continues to drive some of the absorbed water inwards. For components that can be readilly removed from the aircraft it may be possible to overcome the vapour pressure by applying an opposing pressure in an autoclave. However, if this approach is taken it would probably be preferable to limit the area heated during the repair by using a localized heating blanket rather than the autoclave air.

5. DELAMINATION REPAIR

One of the key findings of the previously reported work on delamination repair [1] was that in order to fully infiltrate the tight delaminations typically found in laminates over 5 mm in thickness it was necessary to maintain a high hydrostatic pressure on the resin throughout the entire cure cycle. Consequently, a resin injection device for on-aircraft repair of delaminations should be capable of simultaneously heating and pressurizing the resin. Other factors that were considered in the design of a prototype resin injection device were the ability to inject resin at several points at the same time, i.e. at several discrete locations around a fastener, on a gently curved surface and on both upper and lower wing skins. Figure 5 shows a section through a prototype device that was constructed. A

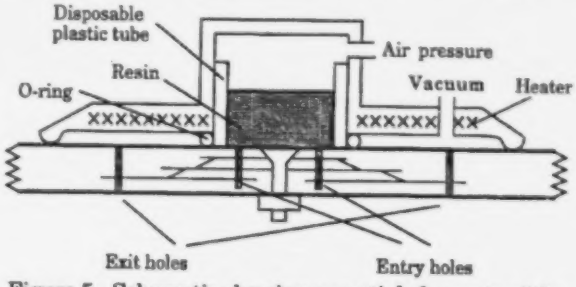

Figure 5 - Schematic showing essential elements of the resin injection device.

vacuum applied to the region between the O-ring and outer seal is used only for holding the device to the aircraft surface. This provides sufficient clamping force to allow the resin contained within the 25 mm diameter O-ring to be injected at pressures of up to 100 psi. A standard hot bonder can be used to provide a source of vacuum and control for the electrical heating element. Fittings (not shown) allow the resin cavity to be evacuated prior to and during introduction of the resin.

A series of tests were undertaken to evaluate the effectiveness of the prototype resin injection device itself and to optimize on-aircraft resin injection procedures. Twelve plates, 20 cm x 20 cm, were cut from a 6 mm thick AS4/3501-6 carbon/epoxy laminate having the following lay-up: $[45_2/-45_2/0_2/45/-45/90_2/45/-45/90_2/45/-45/0_2/45/-45/90/0_3]_s$. Six of the plates were then subjected to low velocity impacts that produced a delaminated region approximately 5 cm in diameter at the centre of each plate. A row of 6.3 mm diameter fastener holes were drilled across each of the remaining six plates and delaminations induced around the centre hole in each plate by loading the countersunk surface of the hole while supporting the back surface against a plate having a circular cut-out. This

P150500.PDF [Page: 6 of 10]

procedure which yielded delaminations up to 4 cm in diameter is described in more detail in reference [1]. Lengths of aluminum angle "iron" were then attached with titanium aerospace fasteners to the back surface of each plate to simulate the wing substructure. All twelve plates were then scanned ultrasonically to produce both amplitude and time-of-flight C-scan images.

Injection repairs were made using a flow-through procedure rather than an evacuate and fill procedure since it is not possible to seal cracks or maintain a vacuum on the inside surface of a wing-skin when carrying out on-aircraft repairs. Because of this, prolonged use of a vacuum is likely to draw air into the delaminations through back surface cracks. Several failed attempts were made before consistently successful repairs were obtained. While the resin injection device behaved more or less as anticipated a trial and error approach had to be taken to optimize the number, size and location of entry and exit holes for infiltrating the resin into the delaminations. The best results were obtained when 6 to 8 entry holes, each 1 mm in diameter were drilled 80% to 90% through the thickness close to the centre of the delaminated region. In the case of the holed laminates the injection holes were placed circumferentially around the head of the fastener. Four 1 mm exit holes equally spaced around the outer portion of the delaminated region were drilled all the way through the laminate, plugged with bleeder cloth to act as a resin dam and then sealed on top to prevent entry of the vacuum. A pressure of 100 psi, a heating rate of 2°C/min and a cure of 2 hours at 140°C were used for most of the repairs. C-scan images of some of the successfully repaired delaminations are shown in Figures 6 and 7.

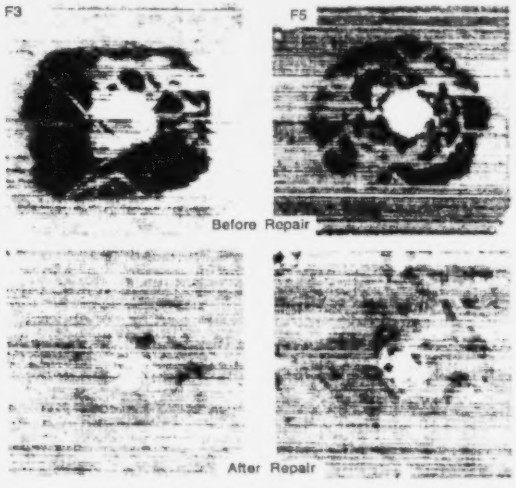

Figure 6 - Ultrasonic C-scan images of repaired fastener hole delaminations.

Although the earlier results had indicated that the C-scans alone gave a good indication of the extent to which the static strength of the repaired laminates was restored, the only fatigue testing that had been done had been limited to characterising the mode II interlaminar fatigue properties. Moreover, injection

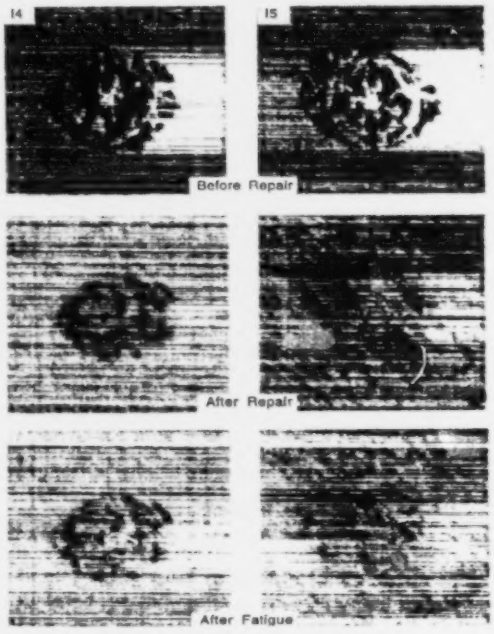

Figure 7 - Ultrasonic C-scan images of repaired impact damaged plates.

repairs tested under a predominantly compressive load spectrum by the US Navy failed at less than 10% of the design life [5]. Two of the repaired impact specimens were therefore evaluated under compression fatigue conditions. The test plates were first trimmed to a size of 100 mm x 150 mm in order to fit into a standard compression after impact (CAI) fixture. They were then subjected to 1 million sinusoidal cycles of constant amplitude compression loading at an R-ratio of 0.07 and a maximum strain of 4000 micro-strain. The plates were then ultrasonically scanned for a final time before being put back into the CAI fixture and loaded to failure in compression. These C-scan images which are shown at the foot of Figure 7 indicate that no damage growth occurred during the fatigue loading. Table 1 lists the failure loads and corresponding strains of the two repaired specimens as well as both an un-repaired and an undamaged specimen.

Test Condition	Failure Load (kN)	Failure Strain (μstrain)
No Damage	306	8740
Impact Only*	151	4310
Impact + Repair	I4 236	6740
	I5 300	8570

* No Fatigue

Table 1 - Post-Fatigue Failure Loads and Strains for Repaired Compression Specimens.

As a final validation of the resin injection repair procedures several demonstration repairs were made to impact damaged regions on an actual aircraft component. One of the outer wings salvaged from a damaged CF-18 aircraft was used for this purpose. One surface of the wing was first inspected ultrasonically to ensure that no non-visible damage was present. A portable drop weight impactor was then used to create delaminated regions at various locations on the wing. The impact energies of around 50 Joules were chosen to produce delaminations approximately 50 mm in diameter but with no back surface damage. Ultrasonic C-scans, similar to that shown in Figure 8, were obtained and actual size images printed out on transparent film. By placing the transparency on the

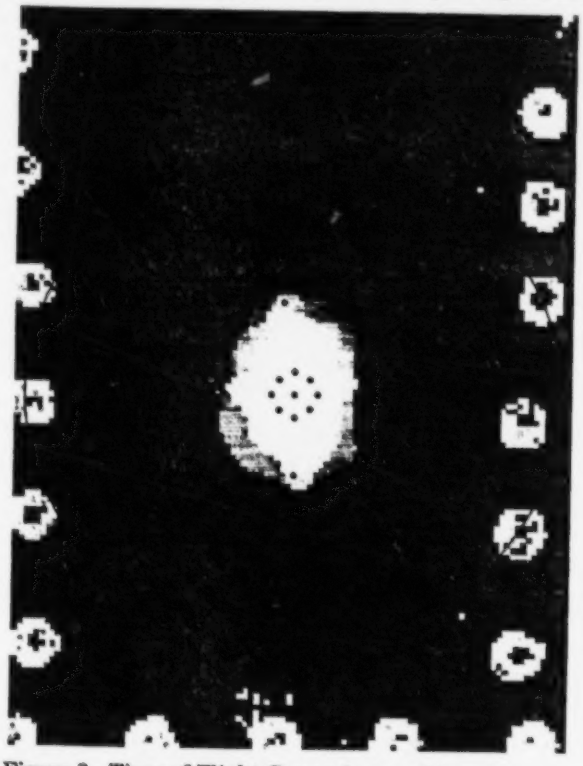

Figure 8 - Time-of-Flight C-scan image of impact damage in CF-18 outer wing.

surface of the wing and using the fastener heads for location purposes entry and exit holes could be drilled into the delaminations in the same manner as described above. Because of the presence of ply-drops, great care had to be exercised in drilling the injection holes close to the back surface without penetrating the laminate. Use was made of the C-scan time-of-flight data, manufacturing drawings of the wing and thickness measurements made through the exit holes. The holes were drilled with a small portable drill mounted to a stand fitted with a micrometer feed. The first repair attempts were once again unsuccessful. This time an immediate problem arose as the resin pressure was being increased. At between 50 and 60 psi the back surface plies delaminated and cracked allowing the resin to leak out. Subsequent repairs carried out at lower injection pressures were moderately successful but better results were obtained by reducing the depth of the injection holes and applying up to the full 100 psi of air pressure. Figure 9 shows examples of the C-scan images that were obtained using both repair

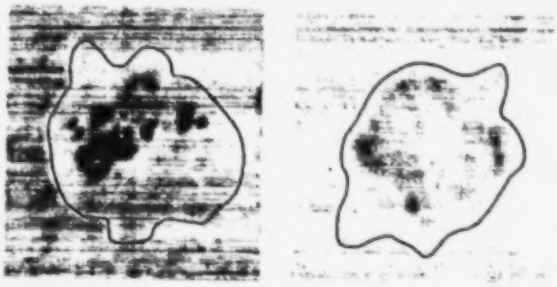

Figure 9 - Ultrasonic C-scans showing in-situ repairs to impact damaged wing skin. (a) reduced pressure. (b) reduced hole depth.

procedures. The original outline of the delaminations are superimposed on the images.

6. BATTLE DAMAGE REPAIR

One of the issues concerning battle damage repair of composites that is often neglected or glossed over involves the cleaning up of through hole damage. From an analytical point of view it is considerably simpler to assess the loss of strength and stiffness and to design a repair if all of the adjacent cracked and delaminated material is removed first so as to form a circular or oval hole. There, are however, two problems with this approach. First, in most ABDR scenarios it is unlikely that non-destructive test support will be available to accurately determine the extent of delamination around a hole. Secondly, depending on the nature of the damage itself, it may be necessary to significantly increase the size of the hole in order to remove all of the delaminated material. While there is a danger that delaminations that are left in place may grow under compressive loads, the ability of the delaminated material to carry tensile and in-plane shear loads is clearly lost once it is removed. In an attempt to try and quantify this trade-off, a series of tests were undertaken to measure and compare the loss in strength and stiffness of panels which had perforation impact damage with those of panels having machined holes.

The test panels, which were made from Hercules AS4/3501-6 carbon-epoxy pre-preg, were 140 mm wide and approximately 3.5 mm thick. Two different laminates were examined, with one having a quasi-isotropic lay-up:

$$[45/-45/0/90/-45/45/0/90/45/-45/0/90]_S$$

and the other an orthotropic stacking sequence more representative of the CF-18:

$$[45/-45/0_2/90/0/-45/0/45/0_2/-45/45/0_2/45/0/-45/0/90/0_2/-45/45]_T.$$

The penetration damage to the panels was carried out at the University of British Columbia, using their high speed gas gun facility [6]. The panels were loosely held against a steel frame having a 100 mm by 200 mm cut-

P130500.PDF [Page: 7 of 10]

As a final validation of the resin injection repair procedures several demonstration repairs were made to impact damaged regions on an actual aircraft component. One of the outer wings salvaged from a damaged CF-18 aircraft was used for this purpose. One surface of the wing was first inspected ultrasonically to ensure that no non-visible damage was present. A portable drop weight impactor was then used to create delaminated regions at various locations on the wing. The impact energies of around 50 Joules were chosen to produce delaminations approximately 50 mm in diameter but with no back surface damage. Ultrasonic C-scans, similar to that shown in Figure 8, were obtained and actual size images printed out on transparent film. By placing the transparency on the

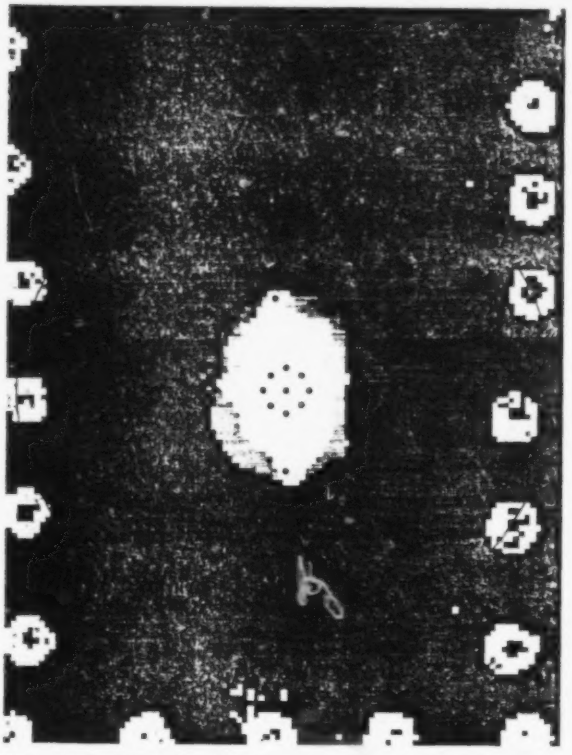

Figure 8 - Time-of-Flight C-scan image of impact damage in CF-18 outer wing.

surface of the wing and using the fastener heads for location purposes entry and exit holes could be drilled into the delaminations in the same manner as described above. Because of the presence of ply-drops, great care had to be exercised in drilling the injection holes close to the back surface without penetrating the laminate. Use was made of the C-scan time-of-flight data, manufacturing drawings of the wing and thickness measurements made through the exit holes. The holes were drilled with a small portable drill mounted to a stand fitted with a micrometer feed. The first repair attempts were once again unsuccessful. This time an immediate problem arose as the resin pressure was being increased. At between 50 and 60 psi the back surface plies delaminated and cracked allowing the resin to leak out. Subsequent repairs carried out at lower injection pressures were moderately successful but better results were obtained by reducing the depth of the injection holes and applying up to the full 100 psi of air pressure. Figure 9 shows examples of the C-scan images that were obtained using both repair

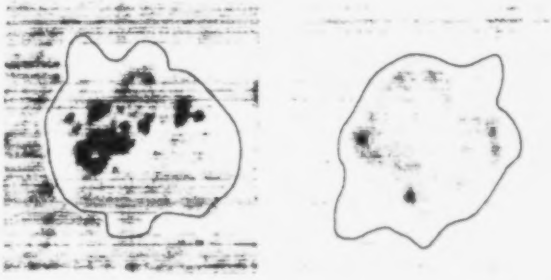

Figure 9 - Ultrasonic C-scans showing in-situ repairs to impact damaged wing skin. (a) reduced pressure. (b) reduced hole depth.

procedures. The original outline of the delaminations are superimposed on the images.

6. BATTLE DAMAGE REPAIR

One of the issues concerning battle damage repair of composites that is often neglected or glossed over involves the cleaning up of through hole damage. From an analytical point of view it is considerably simpler to assess the loss of strength and stiffness and to design a repair if all of the adjacent cracked and delaminated material is removed first so as to form a circular or oval hole. There, are however, two problems with this approach. First, in most ABDR scenarios it is unlikely that non-destructive test support will be available to accurately determine the extent of delamination around a hole. Secondly, depending on the nature of the damage itself, it may be necessary to significantly increase the size of the hole in order to remove all of the delaminated material. While there is a danger that delaminations that are left in place may grow under compressive loads, the ability of the delaminated material to carry tensile and in-plane shear loads is clearly lost once it is removed. In an attempt to try and quantify this trade-off, a series of tests were undertaken to measure and compare the loss in strength and stiffness of panels which had perforation impact damage with those of panels having machined holes.

The test panels, which were made from Hercules AS4/3501-6 carbon-epoxy pre-preg, were 140 mm wide and approximately 3.5 mm thick. Two different laminates were examined, with one having a quasi-isotropic lay-up:

$$[45/-45/0/90/-45/45/0/90/45/-45/0/90]_S$$

and the other an orthotropic stacking sequence more representative of the CF-18:

$$[45/-45/0_2/90/0/-45/0/45/0_2/-45/45/0_2/45/0/-45/0/90/0_2/-45/45]_T.$$

The penetration damage to the panels was carried out at the University of British Columbia, using their high speed gas gun facility [6]. The panels were loosely held against a steel frame having a 100 mm by 200 mm cut-

P150500.PDF [Page: 8 of 10]

out. Three different hemi-spherical tipped impactors having diameters of 7.6, 12.7,and 25 mm and masses of just over 300 grams were fired at the centre of the panels at velocities between 28 and 30 m/sec. Instrumentation on the impactors indicated that the percentage of the incident energy absorbed during penetration was approximately 25% for the 7.6 mm spheres, and between 40% and 50% for the larger impactor sizes. After sealing off the delaminated regions around the holes with tape to prevent water ingress, the panels were ultrasonically C-scanned to determine the extent of the damage. Examples of C-scan images for the quasi-isotropic panels are shown in Figure 10. The size of the holes themselves, as viewed from the front surface, are superimposed on the C-scan images. Similar scans were obtained for the orthotropic panels except that the delamination damage tended to be more elongated in the 0° fibre direction. For the remaining panels, circular holes having diameters ranging from 6.3 to 52 mm were either drilled or cut out using diamond tipped hole saws.

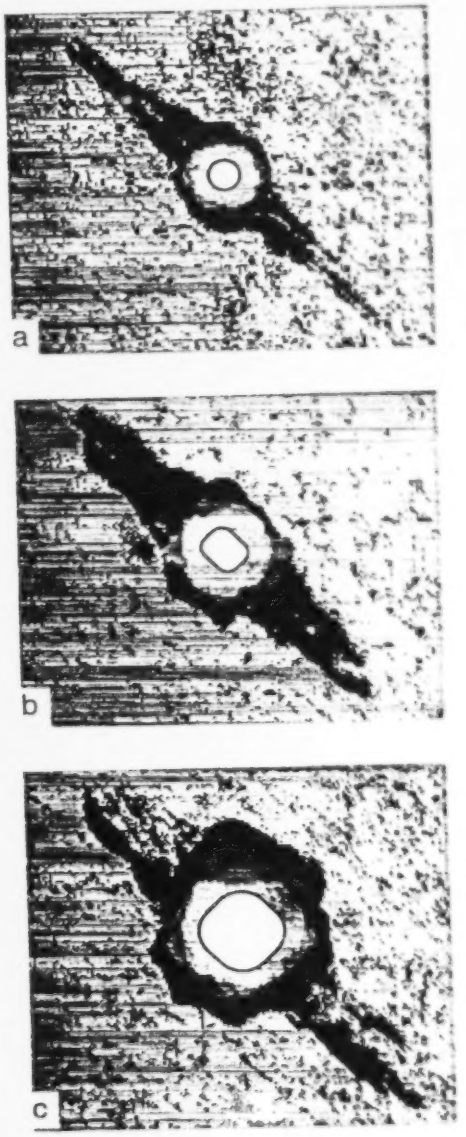

Figure 10 - Ultrasonic time-of-flight images showing delamination around holes in quasi-isotropic panels. (a) 7.6 mm, (b) 12.7 mm and (c) 25 mm impactors.

Mechanical testing of the panels was accomplished in a 1000 kN servo-hydraulic testing machine fitted with hydraulic grips. Anti-buckling guides, which consisted of two plates with adjustable sliding contact points mounted to the columns on either side of the load frame, were used to provide lateral support to the panels during the compression tests. No lateral support was applied to the delaminated regions around the holes. The panels to be tested in tension had an unsupported length of 300 mm while the compressive panels had a gage length of 180 mm. The undamaged tensile strength of each of the laminates was determined independantly using standard 25 mm wide test coupons.

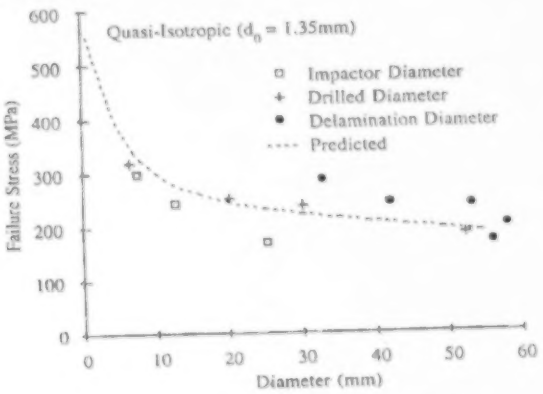

Figure 11 - Tensile Results for the Quasi-Isotropic Panels.

The results for the panels tested in tension are shown in Figure 11 for the quasi-isotropic laminate and in Figure 12 for the orthotropic lay-up. The failure stress of the panels containing the through hole damage is plotted against both the impactor diameter and the delamination diameter as measured perpendicular to the loading direction. The measured performance is compared to predictions based on a point stress failure criterion, which is calculated using the following equation [7, 8] :

$$\sigma_N^{\infty} = \frac{2\sigma_0}{[2+\xi_2^2+3\xi_2^4-(K_t-3)(5\xi_2^6-7\xi_2^8)]}$$

where:

$$\xi_{\gamma 2} = \frac{R}{R+d_0} \quad \text{and} \quad K_t = K_t^{\infty} \cdot \frac{2+\left(1-\frac{2R}{W}\right)^3}{3\left(1-\frac{2R}{W}\right)}$$

P150500.PDF [Page: 9 of 10]

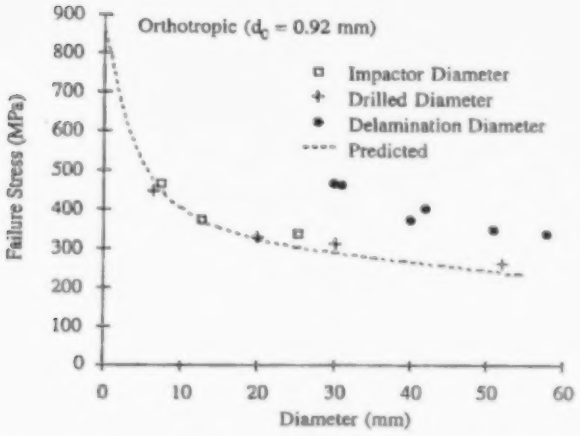

Figure 12 - Tensile Results for the Orthotropic Panels.

The parameter d_o which gave the best fit to the experimental data was found to be 0.92 mm for the orthotropic lay-up and 1.35 mm for the quasi-isotropic. These values are similar to the 1.02 mm reported by Nuismer and Whitney [7,9] for a similar brittle matrix carbon/epoxy composite. It is evident from both of these figures that the point stress failure criteria fits the drilled hole data extremely well. The loss in tensile strength of the orthotropic laminate caused by the perforation damage is almost identical to that of machined holes having the same diameter as the impactor and considerably less than that of machined holes having the same diameter as the delaminations around the hole. In the case of the quasi-isotropic laminate, the loss in strength is greater than that of impactor size machined holes but not as great as it would be if all of the delaminated material were removed. These results are consistent with numerous other investigations that have demonstrated the notch blunting effects of damage (especially fatigue damage) around holes in composite laminates loaded in tension.

Figures 13 and 14 show the results of the compression testing of the quasi-isotropic and orthotropic panels.

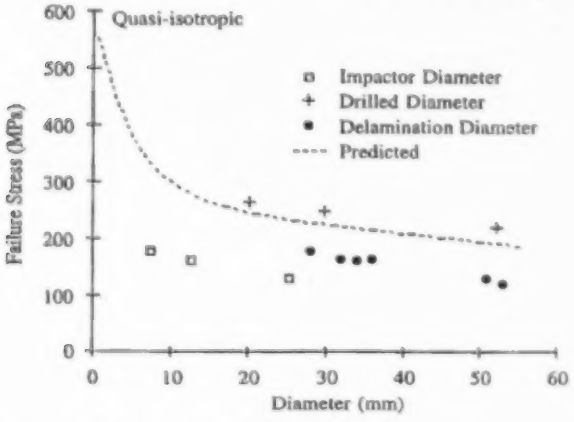

Figure 13 - Compression Results for the Quasi-Isotropic Panels.

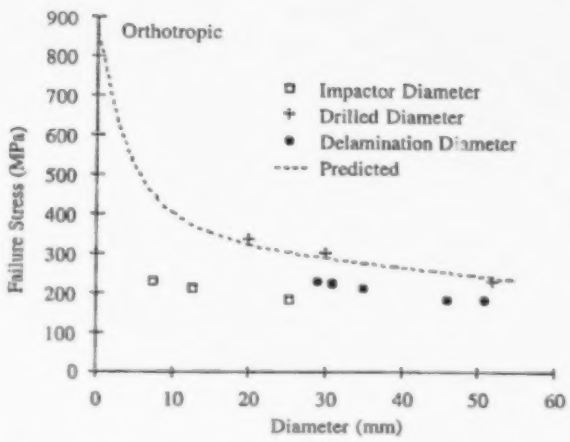

Figure 14 - Compression Results for the Orthotropic Panels.

Here again the point stress predictions and measured failure strengths for the drilled holes are in reasonable agreement. However, the loss in compression strength due to the perforation damage is now greater than that of machined holes having the same diameter as the delaminations. This implies that the delaminated plies are not simply failing to carry load as they buckle outward but that the delaminations are extending towards the panel edges during the failure process. This is supported by a noticeable difference between the two sets of failed panels, in that with the perforated specimens the outer plies had buckled outwards across the entire width without breaking, whereas in the machined hole panels all of the plies had fractured. There would therefore seem to be some potential for reducing the loss in compression strength by stabilizing the delaminated plies and inhibiting delamination growth through the use of blind fasteners to clamp the laminate. Ideally, this could be incorporated into a typical bolted patch repair scheme. Although further investigation is clearly required to verify this concept, previous bolted joint testing [10] which showed that delaminated plies can carry 80% of the shear load of undamaged laminates, suggests that a bolted repair in which fasteners are inserted through both good and damaged material may have some merit.

7. CONCLUDING REMARKS

Investigation of the skin/core debonding encountered during elevated temperature repairs has clearly shown the role of moisture degradation of the aluminum epoxy interface. While several options are available for reducing or eliminating its occurrence during repair, the broader implications of bondline degradation relate more to the long term durability of bonded honeycomb structure. In particular, is the un-treated aluminum core traditionally used with aluminum skins appropriate for use with composite skins where moisture will eventually reach the bondline by diffusion?

The work on delamination repair reported here has shown that resin injection can be an effective and practical means of restoring the strength of impact damaged composite components. With further

P150500.PDF [Page: 10 of 10]

development work it should even prove possible to devise the necessary jigs and modifications to allow repairs on the underside of a wing or on a steeply sloping vertical stabilizer skin. Nevertheless, the degree of skill required to implement repairs of this type as well as the need for an ultrasonic C-scan capability will limit its use to a third line (ie. depot level) repair facility.

The experiments to assess the need to remove delaminations associated with battle damage holes have demonstrated that the penalty in terms of tensile strength is at least as great as the gain in compressive strength. When the constraints of carrying out repairs in an ABDR environment are taken into account it would appear that the best approach may be to explore "blind" repair concepts which combine ply stabilization with load transfer into a bolted patch.

8. ACKNOWLEDGEMENT

The authors wish to acknowledge the following individuals for their valuable contributions to the work presented in this paper: Mr. D Lindahl of LCM Corp., Vancouver, BC for development of the resin injection device, Dr. D. Delfosse of the University of British Columbia, Metals and Materials Department for the high velocity impact testing and Mr. E. Jensen and Mr. T. Miller of DREP for their assistance in carrying out most of the other experimental tasks.

9. REFERENCES

[1] Russell, A.J. and Bowers, C.P., "Repairing Delaminations with Low Viscosity Epoxy Resin", AGARD-CP-530, 1992, p.27-1

[2] Perl, D., Private communication, Naval Air Depot, San Diego, 1993.

[3] Venables, J.D., McNamara, D.K., Chen, J.M., Ditchek, B.M., Morgenthaler, T.I., Sun, T.S. and Hopping, R.L., "Effect of Moisture on Adhesively Bonded Aluminum Structures", *Proceedings of 12th National SAMPE Technical Conference*, 1980, p. 909-923

[4] Alwitt, R.S., "The Aluminum-Water System", in *Oxides and Oxide Films*, Vol 4, Diggle, J.W. and Vijh, A.K., eds., Marcel Dekker, New York, 1976, Chap 3.

[5] Ostrom, R.B., Stone, R.H., Fogg, L.O. and Smith, L.W., "Field Level Repair for Composite Structures", NADC Report No. 79174-60, 1985.

[6] Delfosse, D., Pageau, G., Bennet, R. and Poursartip, A., "Instrumented Impact Testing at High Velocities", Journal of Composites Technology & Research, Vol. 15, No. 1, 1993, p. 38-45.

[7] Nuismer, R.J. and Whitney, J.M., "Uniaxial Failure of Composite Laminates Containing Stress Concentrations", *Fracture Mechanics of Composites, ASTM STP 593*, American Society for Testing and Materials, 1975, p. 117-142.

[8] Peterson, R.E., *Stress Concentration Factors*, Wiley-Interscience, 1974, p.110-111.

[9] Whitney, J.M. and Nuismer, R.J., "Stress Fracture Criteria for Laminated Composites Containing Stress Concentrations", Journal of Composite Materials, Vol. 8, July 1974, p. 253 - 265.

[10] Russell, A. J., and McDougall, G., "A Preliminary Evaluation of Blind Fasteners for Battle Damage Repair of CF-18 Graphite/Epoxy Components", DREP Technical Memorandum 92-30, September 1992.